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AT SUPERSONIC AND HYPERSONIC MACH NUMBERS

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National Advisory Committee for Aeronautics
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INTRODUCTION

Research dealing with the problem of aerodynamic heating of high-speed missiles and airplanes has occupied a prominent place in aerodynamic interests in the past ten years and will continue to grow in importance as speeds continue to increase. Because of the limited understanding of many of the physical processes involved, considerable dependence must be placed on experiment as a source of information to guide theory and design. Accordingly, there has been activity along several lines of attack in the facilities of the NACA to discover and apply suitable methods for studying aerodynamic heating experimentally. The problem has several aspects. It is difficult, first of all, to simulate adequately the heat-transfer conditions which occur during flight through the atmosphere because a number of simulation variables in addition to those normally used in aerodynamic investigations enter the problem. Also, it is frequently found that the methods used to achieve simulation make the problems of measurement difficult. It will be the purpose of this paper to review the experience gained at the Ames Laboratory of the NACA with respect to the simulation and measurement of aerodynamic heat transfer at supersonic and hypersonic speeds.

SYMBOLS

c_p heat capacity at constant pressure, Btu/slug × °R

$\dfrac{C_F}{C_{F_i}}$ ratio of average skin-friction coefficient to that at zero Mach number and zero heat transfer for the same length Reynolds number, dimensionless

$\dfrac{c_f}{c_{f_i}}$ ratio of local skin-friction coefficient to that at zero Mach number and zero heat transfer for the same length Reynolds number, dimensionless

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Subscript

o boundary-layer-edge conditions

SIMULATION

As in all aerodynamic testing, similarity parameters are used as a guide in the design of aerodynamic heating experiments. In contrast to conventional force and stability testing, however, where Mach number and Reynolds number are the primary similarity parameters, it is necessary in aerodynamic heating tests to give attention to additional parameters. This is required because aerodynamic heating is a boundary-layer phenomenon, and the properties of the boundary layer are determined by temperature as well as by elastic, viscous, and inertia forces. Consequently, we should expect to encounter dimensionless parameters whose values depend upon temperature.

In discussing simulation herein, emphasis will be placed on recognizing the significant variables in the problem. The dependent variable is the dimensionless heat-transfer coefficient, the Stanton number, defined in equation (1).

$$St = \frac{h}{c_p \rho u} \tag{1}$$

The Stanton number is related to the heat-transfer rate as shown in equation (2).

$$q = h(T_r - T_w) = St\ c_p \rho u(T_r - T_w) \tag{2}$$

It should be noted that the temperature of the body surface and the boundary-layer recovery temperature are essential to definition of the heat-transfer rate. For a given value of the Stanton number, heat-transfer rate can vary from large positive values through zero to large negative values, depending on body-surface and recovery temperatures. This is mentioned to clarify the definition and significance of the Stanton number which may be regarded as a dimensionless factor of proportionality between heat-transfer rate and the temperature potential causing heat transfer. It will be presumed that in a given problem, the body-surface temperature and the stream properties are known. What is required, then, is information on the recovery temperature and Stanton number.

The variables which are known to affect Stanton number are shown in equation (3).

$$St = f\left(M,\ R,\ Pr,\ \frac{T_w}{T_o},\ R_t,\ T_o,\ \frac{dp}{dx},\ \frac{dT_w}{dx},\ \frac{\lambda}{\delta}\right) \qquad (3)$$

The Mach number and Reynolds number are rather naturally expected to affect the heat-transfer problem as they do other aspects of aerodynamics in general and boundary-layer flow in particular. The Prandtl number can be expected to have importance because it represents the thermal properties, conductivity and heat capacity, of the fluid. The additional variables require some comment.

The body-surface temperature to boundary-layer-edge temperature ratio, T_w/T_o, enters the problem as it affects the temperature and density structure of the boundary layer. In figure 2, there are shown the differences which occur in boundary-layer density and temperature profiles when the wall temperature is varied from a relatively cold value, $T_w = T_o$, to a hot value, $T_w = T_r$. These data are for laminar boundary layer and were taken from the calculations of Klunker and McLean, reference 1. Not only are the temperature and density distributions grossly altered by the change in wall temperature, but the boundary-layer thickness is affected as well. With changes of this magnitude in the boundary-layer structure, it should not be expected or assumed that the Stanton number will be unaffected. Recent data on the skin friction of turbulent boundary layers (refs. 2, 3, and 4) relate to this effect and are shown in figure 3. The data show appreciable differences in the measured skin friction at a given Mach and Reynolds number between the tests with low wall temperature and those with the wall temperature near the boundary-layer-recovery temperature. Since close relationship exists between the dimensionless coefficients of skin friction and heat transfer, references 5 and 6, similar effects are expected in the case of heat transfer. This implies that the Stanton number and heat-transfer coefficient, h, will not be constant over wide ranges of body-surface temperature.

The variable R_t in figure 1 is the transition Reynolds number and is important because of the large differences in the heat-transfer characteristics of laminar and turbulent boundary layers. This variable is included to emphasize the importance of knowledge of the flow type in experimental investigation as well as in flight. An otherwise valid experiment conducted with a transitional boundary layer instead of a laminar or turbulent boundary layer can easily become misleading and valueless. The transition Reynolds number, it is felt, should be considered an independent variable in the heat-transfer problem. Also, the need for valid information on transition Reynolds number for the flight condition is apparent.

The free-stream air temperature, T_O, is used in equation (3) as an index of all the air temperatures in the problem. Fixing T_O, along with Mach number and wall temperature, fixes the temperature level throughout the boundary layer and flow field. The importance of temperature level at hypersonic speeds is that it determines the occurrence of such effects as the activation of vibrational heat capacity in the fluid, or the occurrence of molecular dissociation, whereby some of the normally diatomic molecules of oxygen and nitrogen absorb sufficient vibrational energy to separate into a monatomic state. Since these effects depend on the temperature level of the air, their simulation requires that flight temperature levels, that is, free-stream temperatures from 400° to 600° Rankine and stagnation temperatures of many thousands of degrees Rankine, depending on the Mach number, be simulated.

One of the principal questions relative to dissociation is the time required for it to occur. That is, assuming that the temperature levels are such that, in time, appreciable dissociation will occur, then will it occur fast enough to make itself felt in the short time available as the air streams through the high temperature flow field of the body? This suggests that still another variable is needed for simulation, the time for dissociation compared to the time of influence of the body flow field, $u\tau/l$. The importance of this variable with respect to heat-capacity lag was recognized by Kantrowitz in reference 7. A complicating factor is that the time for dissociation depends on the gas composition and pressure. The presence of water vapor, for example, changes significantly the time for the vibrational heat capacities to reach equilibrium, reference 8. Therefore, it appears that flights under varying conditions of humidity will not experience the same dissociation and vibration effects, and the condition simulated will depend on the humidity in the test stream.

A secondary effect of free-stream temperature is related to the laws governing the variation of thermal conductivity and viscosity with temperature. Temperature varies across the boundary layer and causes these air properties to vary. However, the laws of variation in the very low temperature region differ from those at higher temperature. Hence, a full simulation of this effect will require simulation of flight temperature levels.

The variables dp/dx and dT_W/dx express the influence of body shape and surface-temperature gradient. As such, they are beyond the scope of the present discussion which is confined to facility effects primarily. Needless to say, they will influence the heat-transfer problem and must be simulated.

For the case of aerodynamic heating occurring at very high altitudes, parameters based on molecular properties such as the Knudsen number (ratio of mean free path to characteristic dimension) and molecular speed ratio (ratio of flight speed to the most probable molecular speed) are encountered, references 9 and 10. It is interesting to note that these parameters can be related to Mach number and Reynolds number so that actually, no new parameters are involved. It is important, however, that heat-transfer laws derived for continuum flow not be applied under conditions of slip flow or free molecule flow - conditions whose boundaries can be expressed in terms of the Knudsen number.

These are the variables which influence aerodynamic heating and which must therefore be simulated. It is a formidable list. What, then, are the methods available for achieving this simulation?

The Problem of Simulation in Wind Tunnels

The problem of simulating flight conditions in a wind-tunnel test designed to measure convective heat transfer to a specific body reduces to the simulation of the correct air temperature, body temperature, Mach number, and Reynolds number. Simulation of the correct air temperature is probably the most difficult task for moderate Mach and Reynolds numbers. Figure 4 shows the stagnation air temperature required in a supersonic wind tunnel to maintain a test-section static temperature of 400° Rankine. It would appear from this figure that continuous wind tunnels, which are capable of simultaneously simulating flight temperatures and Mach numbers are probably not feasible above a Mach number of 4 or 5. In a high Mach number, high stagnation temperature wind tunnel, difficulty is also experienced in attaining high Reynolds numbers. Figure 5 shows the stagnation pressure which must be used in order to maintain several Reynolds numbers while sumultaneously simulating flight temperatures. The very high pressures required to fully simulate flight conditions even at a Mach number of 6 and a Reynolds number of 5 million per foot will obviously cause severe problems.

One wind tunnel, designed primarily for heat-transfer studies, which has recently been completed, is the 10- by 12-inch heat-transfer wind tunnel located at the Ames Laboratory. This wind tunnel operates at Mach numbers from 2.5 to 6 and is capable of simulating the correct static temperature up to Mach numbers of at least 4. This requires a stagnation temperature of 1660° Rankine. The stagnation pressure may be varied from less than 1 to a maximum of 11 atmospheres which gives a maximum Reynolds number of 14 million per foot. This Reynolds number, however, is not obtained at the highest stagnation temperature. At the highest stagnation temperature at a Mach number of 4, the maximum Reynolds number is about 3 million per foot. At still higher Mach numbers the correct air temperature is not simulated completely. Hence, the degree of simulation decreases as the Mach number increases.

The test section of this wind tunnel is composed entirely of stainless steel which is required from the standpoints of strength and resistance to oxidation. The wind tunnel is essentially uncooled. This feature permits the correct recovery temperature on models to be obtained without having to apply a correction for radiant heat loss to the walls. The nozzle blocks are rigid but movable by means of jacks. The side walls are held by air-loaded jacks and are moved out when the nozzle blocks are moved to change the Mach number. After the nozzle blocks have completed their motion, the side walls are clamped tightly against the nozzle blocks by means of these air-loaded jacks, thus effecting a metal-to-metal air seal.

Beyond a Mach number of 4, the difficulty of obtaining temperature simulation by use of a wind tunnel with a heated air supply increases and it becomes imposingly difficult at Mach numbers greater than 6. There have been devised, therefore, some wind tunnels of specialized type, designed to operate at high temperature levels for short duration. The Ames hypersonic gun tunnel is in this category. It consists of a conical nozzle attached to the muzzle end of a gun. A nylon piston is driven down the bore of this gun by means of a powder charge placed behind it. As the nylon piston progresses, it compresses the air in front of it. It travels at such a speed that shock waves are formed which proceed ahead of the piston to the end of the barrel and reflect repeatedly, thus heating the air in a highly nonisentropic fashion. The piston is brought to rest with some oscillation when the pressure of the compressed air becomes equal to the pressure of the expanded powder gases. A valve is then opened to permit the air to flow out of the gun barrel through the small conical nozzle in front of which is placed the model to be tested. The time for the compression stroke of the nylon piston is only a fraction of a second, but the duration of the test air flow is then about 1 second. Stagnation temperatures of about 2500° Rankine are obtained at Mach numbers of approximately 10. This stagnation temperature, while high by usual standards, is still considerably below the flight value at this Mach number. Nevertheless, this facility is useful for studying heat-transfer problems at high temperature levels and hypersonic speeds.

For investigation of certain features of hypersonic heat transfer, it is possible to use substitute gases instead of air in steady-state wind-tunnel tests. For example, the use of helium is advantageous because it will avert the problem of fluid condensation at high Mach numbers. Of course, some of the simulation variables are not satisfied in tests with substitute gases, notably the temperature level and the wall to free-stream temperature ratio, but this does not preclude the useful application of such facilities to certain problems. For example, the effects of pressure and surface-temperature gradients at hypersonic Mach number, the problem of shock-wave boundary-layer interaction near the leading edge, and the variation of Stanton number with Mach number in dissociation-free flow with small heat transfer can be investigated, to name a few. At Ames Laboratory, a 1- by 10-inch helium channel is under development. Helium is supplied to the stagnation chamber of the nozzle at a pressure of approximately 70 atmospheres and at about room temperature. The Reynolds numbers attained in this wind tunnel are high even at high Mach numbers. This is due primarily to the fact that the density does not decrease with rising Mach number as fast in helium as in air because of the difference in γ. The Reynolds number per foot in the Ames facility will be 7.5 million at a Mach number of 20 when the stagnation pressure is 68 atmospheres. The Mach number range projected is from low supersonic speeds to a Mach number of 20.

In reference 11, Chapman has proposed the use of additional substitute gases, mixtures of heavy polyatomic and heavy monatomic gases. The primary reason for the use of these gases is the relatively low speed of sound which occurs in heavy gases. This permits testing at higher Mach numbers for a given velocity. The use of heavy gas mixtures has the advantage over the previously mentioned helium nozzle in that the correct value of γ can be obtained and, in fact, the variation of γ with Mach number can be made to match that which occurs in air during flight. The disadvantage in the use of heavy gases is that the dissociation which occurs in air is not simulated, although substitute gases have been used in fundamental studies of dissociation phenomena.

Simulation in Free-Flight Ranges

For more complete simulation of flight conditions at high supersonic speeds, particularly with respect to temperature, it would appear necessary to use something other than a wind-tunnel technique. One possibility is free-flight testing at small scale to develop the very high stagnation temperatures and boundary-layer temperatures by the same aerodynamic actions as occur in full-scale flight. This technique has been applied in a facility at the Ames Laboratory referred to as the supersonic free-flight wind tunnel (ref. 12). Models are gun-launched in this facility at high speeds and, to promote the attainment of high supersonic Mach number, are flown upstream through the test section of a supersonic wind tunnel with an air-stream Mach number of 2 so that the velocity of the air adds to that of the model. The air in the test section has a temperature of approximately 300° Rankine due to expansion from an unheated reservoir to a Mach number of 2. This makes it possible to achieve higher test Mach numbers than would otherwise be possible because of the lowering of the speed of sound. However, it also brings the temperature level of the tests below the desired range for full simulation of atmospheric flight. At a Mach number of 10, for example, the stagnation temperature without dissociation for flight at altitudes between 30,000 and 100,000 feet is computed to be 8400° Rankine whereas in this facility it would be three-fourths as great or 6300° Rankine. Thus, test stagnation temperatures usefully close to the flight stagnation temperatures are realized. By slight modification the facility could be made to simulate fully flight air temperature levels. Substitution of a Mach number 1.3 nozzle for the existing Mach number 2 nozzle would produce this result, as would preheating the supply air to a temperature of 720° Rankine or 260° Fahrenheit. Tests can also be made using still air as the test medium to obtain results be applied, recovery temperature can be measured indirectly by observing the variation in heat flow rate as a function of body temperature. Extrapolation of the data to obtain the body surface temperature for zero heat flow yields, by definition, the recovery temperature. However, caution must be used in applying this procedure since, in cases where the heat-transfer coefficient is not constant with temperature difference, the extrapolation will not be linear.

Measurement of the recovery factor, r, requires, in addition to measurement of the recovery temperature, knowledge of the Mach number and static air temperature. The static temperature is usually obtained by computation from measurements of the stagnation temperature. The latter is usually measured directly in the settling chamber of the wind tunnel where the air velocity is low enough that it can be assumed that the air temperature is the stagnation value. The primary precaution that must be observed is that a true average temperature is measured. Usually, the stagnation temperature is obtained by averaging the readings from a number of thermocouples placed throughout the settling chamber. The other quantity required, the Mach number, is computed from measurements of the static and pitot pressures in the test section.

The measurement of the heat flow rate is probably the most difficult measurement that must be made in a heat-transfer experiment. By far, the most accurate method is direct measurement of electrical heat input. In order to see how this is accomplished, let us examine the model shown in figure 6(a). This model which is similar to one used in reference 13 is composed of several heating sections formed by individual heating elements set flush with the surface. Use of separate heating sections allows the heat input to each to be adjusted so that any desired surface temperature distribution can be obtained. Direct measurement of the electrical heat input to these sections at thermal equilibrium then allows the heat-transfer rate to each section to be determined directly. Another method which has been used at the Ames Laboratory (ref. 14) is to use the body shell itself as the resistance element as shown in figure 6(b). In this technique a heavy electrical current is introduced along the central shaft and passed through the body shell. The body resistance then dissipates the power and from the total current flowing and the local voltage drop along the body, the local heat input to the body can be determined. Despite its accuracy, the electrical method has the disadvantage that it must be used essentially under steady-state conditions so that it cannot be applied in tests of too short duration. Also the direction of heat flow is reversed from the most interesting and practical case; that is, the heat flow is out of the body instead of into the body.

There are two methods which can be used to measure heat flow into bodies from an air stream. The first is the use of heat meters. This technique has been well described in reference 15. The heat meter is a thin slab of thermally conducting material which has attached to its opposite faces the hot and cold junctions of a thermopile to measure the temperature drop across the slab. The heat meter is inset into the body under observation. By the laws of steady-state heat conduction, the temperature drop across the slab is a measure of the amount of heat flowing through it.

The second method which lends itself to the measurement of heat-transfer rates of either heated or cooled bodies is the transient technique. This method also has the virtue that it can be used in transient wind tunnels, such as the hypersonic gun tunnel, shock tubes, or free-flight ranges as well as in continuous wind tunnels. In this technique, the body surface temperature is made different from the air-stream recovery temperature by heating or cooling prior to measurement. Then, in the test, the body is allowed to approach the recovery temperature of the stream under the action of the air flow. The time rate of change of body temperature is a measure of the instantaneous rate of heat transfer. The rate of change of temperature with time of each element of volume must be measured, and the mass and heat capacity of each element must be known. This has been proven a very useful technique and has been used by many investigators (e.g., refs. 16 and 17).

Free-Flight Range Measurements

Methods for measuring aerodynamic heating in small-scale free-flight tests are under development at the present time. Techniques have been perfected for studying the related topics of boundary-layer transition (ref. 18) and skin friction (ref. 2). In the remainder of this paper, the methods which are now under study for obtaining heat-transfer data directly will be reviewed.

In this type of test, optical methods of measurement are to be preferred since optical data on the flight are relatively easy to obtain in the form of spark shadowgraphs or short-duration pictures of other kinds. Accordingly, the possibility of measuring heat transfer by interferometry is being explored at the present time. A Mach-Zehnder interferometer with a working field of 10- by 14-inches is used to record a spark interferogram of 3 microseconds duration of the test model in flight. One of the pictures which has been obtained is shown in figure 7 and a corresponding shadowgraph of a similar test model is shown in figure 8. As is evident from the shadowgraph, the boundary layer in this case was turbulent. The boundary layer is also visible in the fringe pattern. The fringe shifts give a quantitative measure of the boundary-layer density distribution. Errors due to refraction of light in the boundary layer appear small for the particular conditions of these tests. Reduction of the fringe-shift data to density data proceeds from the assumption of axial symmetry. The boundary layer is divided, for purposes of computation, into a finite number of annular elements and it is assumed that the density varies linearly through each. Density data obtained in this way are converted to temperature profiles by assuming that the static pressure in the boundary layer is independent of distance from the surface and therefore that the product, ρT, is constant. Temperature distributions obtained in this way are shown in figure 9. An appreciable variation in profiles is obtained, not all of which is due to experimental scatter. In large part, the variations are real, due to the unsteady character of the turbulent boundary layer. Recent pictures obtained under more favorable experimental conditions show distinct variations in the fringe contour of adjacent fringes indicating the unsteadiness of the boundary layer. To obtain a time average profile, therefore, it is necessary to average the profiles recorded.

If the slope of the temperature profile at the wall, $(dT/dy)_w$, could be accurately determined from these data, then it would provide directly a measure of the heat transfer to the wall. Unfortunately, this procedure cannot be used because the large temperature gradients near the wall cannot be adequately resolved from the interferogram. Therefore, an alternate way of determining heat transfer from the data was sought. It was felt that an energy balance method, analogous to the momentum method of determining skin friction from density and velocity profiles, would be suitable and would offer the advantage of depending on the measured profiles throughout the boundary layer rather than depending entirely on values close to the wall. An energy balance is written using either the free-flight frame of reference or the wind-tunnel frame of reference. In the wind-tunnel case (model considered fixed and air in motion) the kinetic energy lost by air in the boundary layer under the action of the skin-friction force is set equal to the sum of the thermal energies gained by the boundary layer and the model. The kinetic energy and thermal energy changes in the boundary layer are evaluated from the profile data and from skin-friction data with the assumption that the velocity profile is of the power law form, $u/u_o = (y/\delta)^n$. The exponent n is evaluated from skin-friction data and density-profile data by use of the momentum equation. This technique appears promising for cases where the boundary-layer thickness is sufficient to permit accurate fringe measurements.

Another method of measurement which is being considered is to insert fusible rings into the surface of a test model. These rings are inset in shallow circumferential grooves a few thousandths of an inch deep and are then polished smooth and fair so that under normal circumstances no shock wave is produced at the insert. Should the model temperature at the location of the fusible ring rise to the melting point of the ring, then melting will begin and a shock wave will be perceived in shadowgraph pictures. Location of the flight distance required to initiate melting will give a measure of heat-transfer rate. This method will require the use of elevated pressure installations having greater length than those now in service so as to permit temperature rise of sufficient magnitude to occur.

In still other cases, the desired information does not require so quantitative a treatment. For example, in the case of projectiles moving at meteoric velocity, say 12,000 feet per second or higher, there are questions as to whether a given surface will spall, vaporize, burn, or melt and flow, or in just what way it will respond to aerodynamic heating. Of course, considerable information of this kind can be obtained directly from the shadowgraph pictures themselves. These are some of the measurement techniques which are under current consideration. It is anticipated that additional ones will be devised as work with these facilities progresses.

SUMMARY

In summary, then, the NACA has been applying a number of different methods to the experimental study of aerodynamic heating at supersonic and hypersonic Mach numbers. The problem is complicated by the many additional similarity variables to be observed in addition to Mach number and Reynolds number. Notably, the temperature level of the experiments becomes of importance. In order to attain the temperature conditions of flight, it is possible at moderate supersonic Mach numbers to heat the air supply of a wind tunnel as is done in the Ames 10- by 12-inch heat-transfer wind tunnel. At higher speeds more novel wind-tunnel techniques are required to produce for short periods the desired high-temperature conditions. A facility in this category is the Ames hypersonic gun tunnel. At still higher speeds conventional wind tunnels are unsuitable for the attainment of extremely high stagnation temperatures such as will be encountered in flight at Mach numbers of the order of 10. A method which has been used to simulate flows of this type has been small-scale free-flight testing. Measurement techniques have been developed for measuring quantities closely related to aerodynamic heating in this type facility, and methods for directly investigating aerodynamic heating and its effects are currently under development.

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$$St = \frac{h}{C_p \rho \mu} \tag{1}$$

$$q = h (T_r - T_w) = St \; C_p \rho \mu (T_r - T_w) \tag{2}$$

$$St = f\left(M, R, Pr, T_w / T_o, R_t, T_o, \frac{dp}{dx}, \frac{dT_w}{dx}, \frac{\lambda}{\delta}\right) \tag{3}$$

$$T_r = T_o \left(1 + r \frac{\gamma - 1}{2} M^2\right) \tag{4}$$

Figure 1

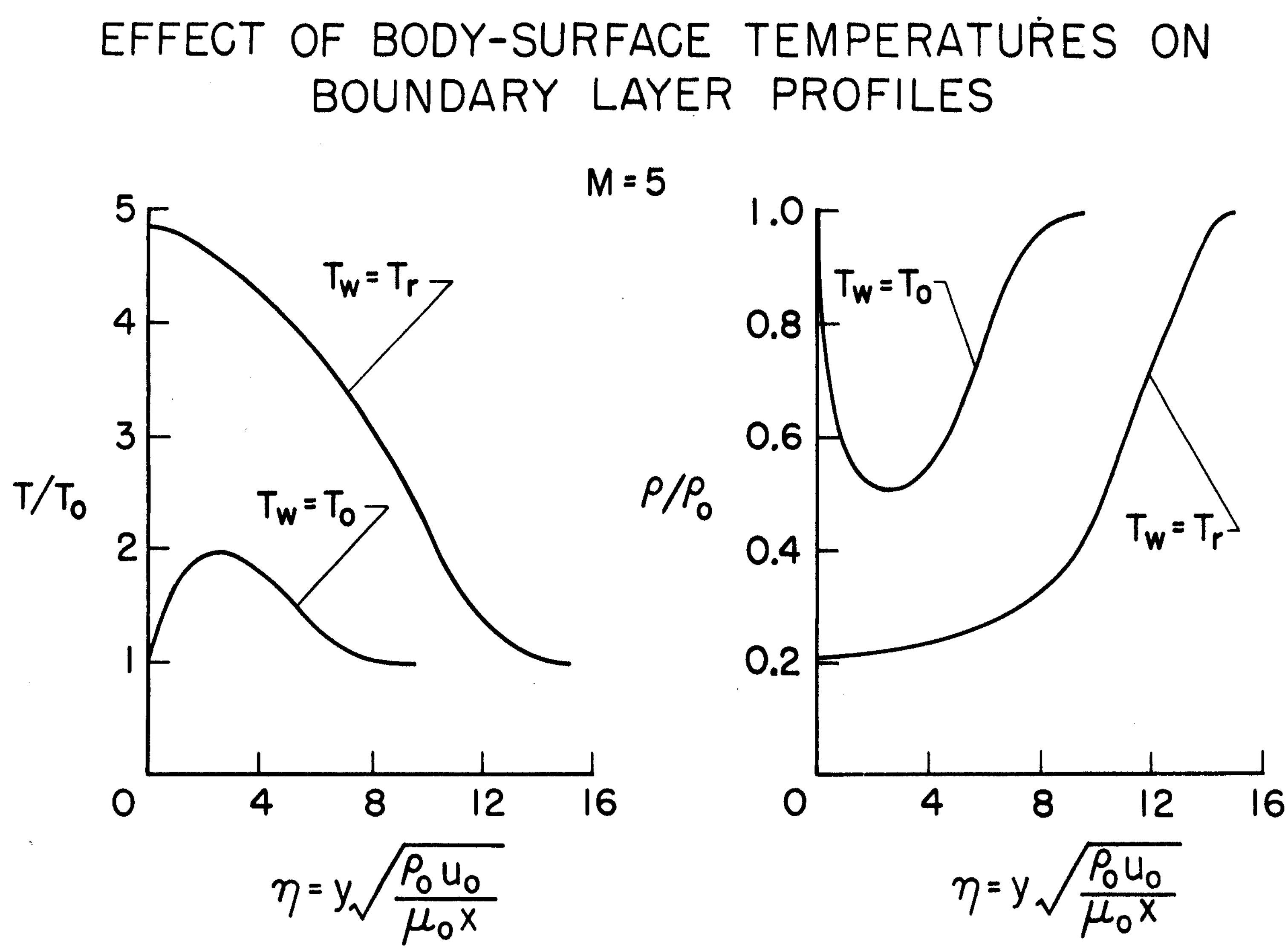
EFFECT OF BODY-SURFACE TEMPERATURES ON
BOUNDARY LAYER PROFILES
M = 5
T/To
Tw = Tr
Tw = To
ρ/ρo
0
4
8
12
16
5
4
3
2
1
1.0
0.8
0.6
0.4
0.2
$\eta = y\sqrt{\frac{\rho_0 u_0}{\mu_0 x}}$

Figure 2

EFFECT OF BODY-SURFACE TEMPERATURE ON TURBULENT SKIN FRICTION

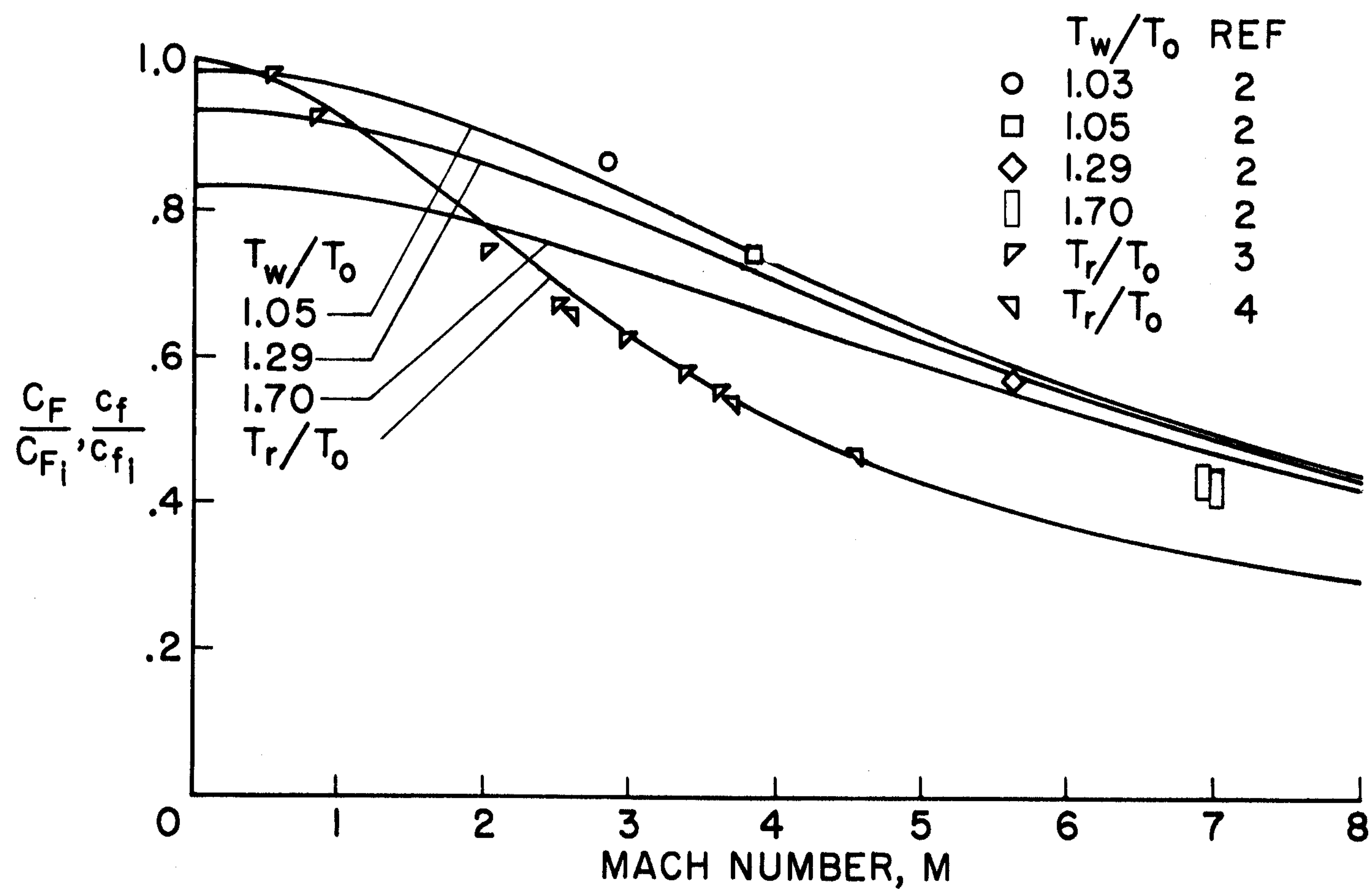

Figure 3

STAGNATION TEMPERATURE REQUIRED IN WIND TUNNELS TO MAINTAIN STREAM STATIC TEMPERATURE OF 400° R

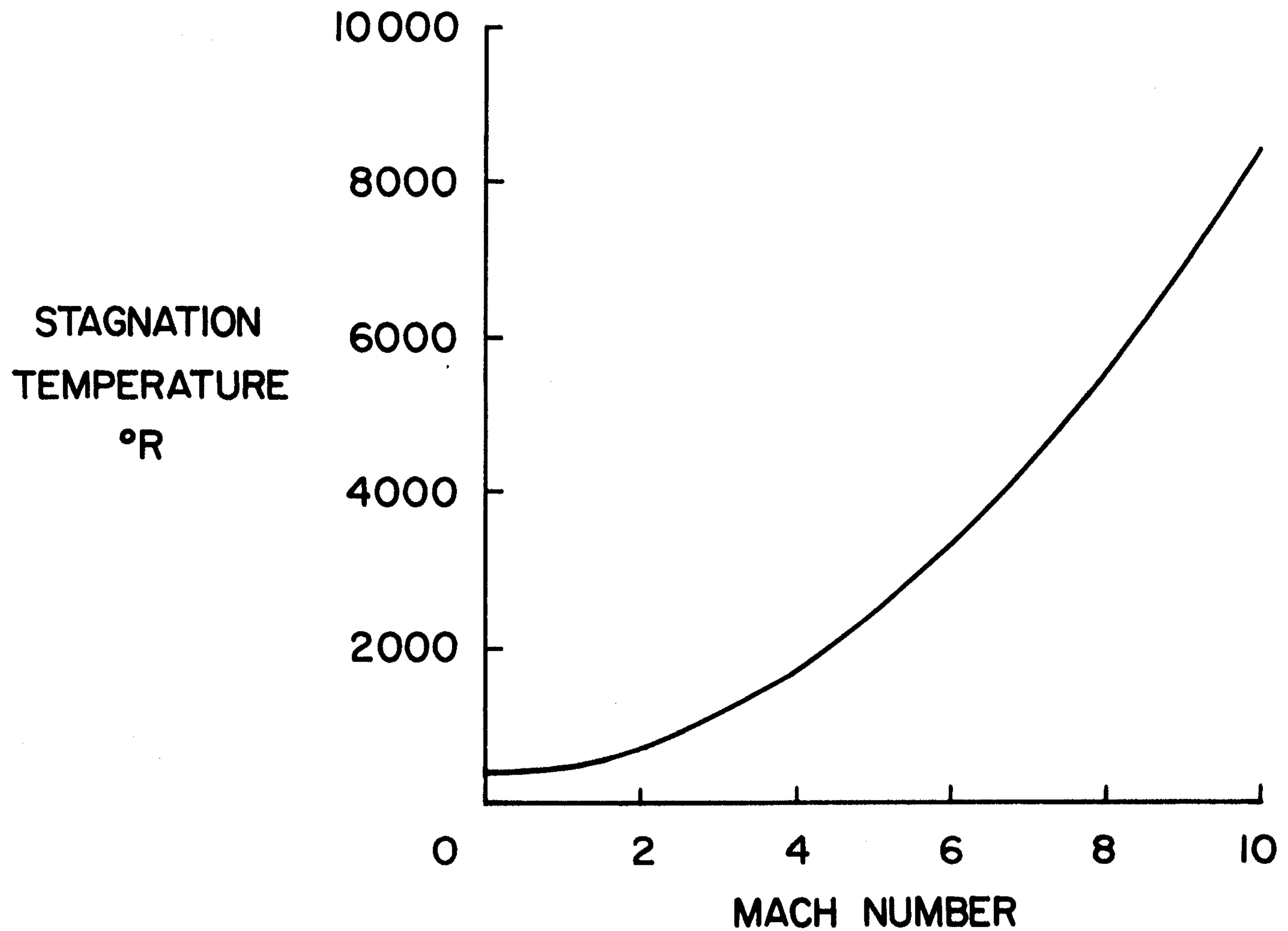

Figure 4

STAGNATION PRESSURE REQUIRED TO SIMULATE FLIGHT REYNOLDS NUMBERS AND STAGNATION TEMPERATURES

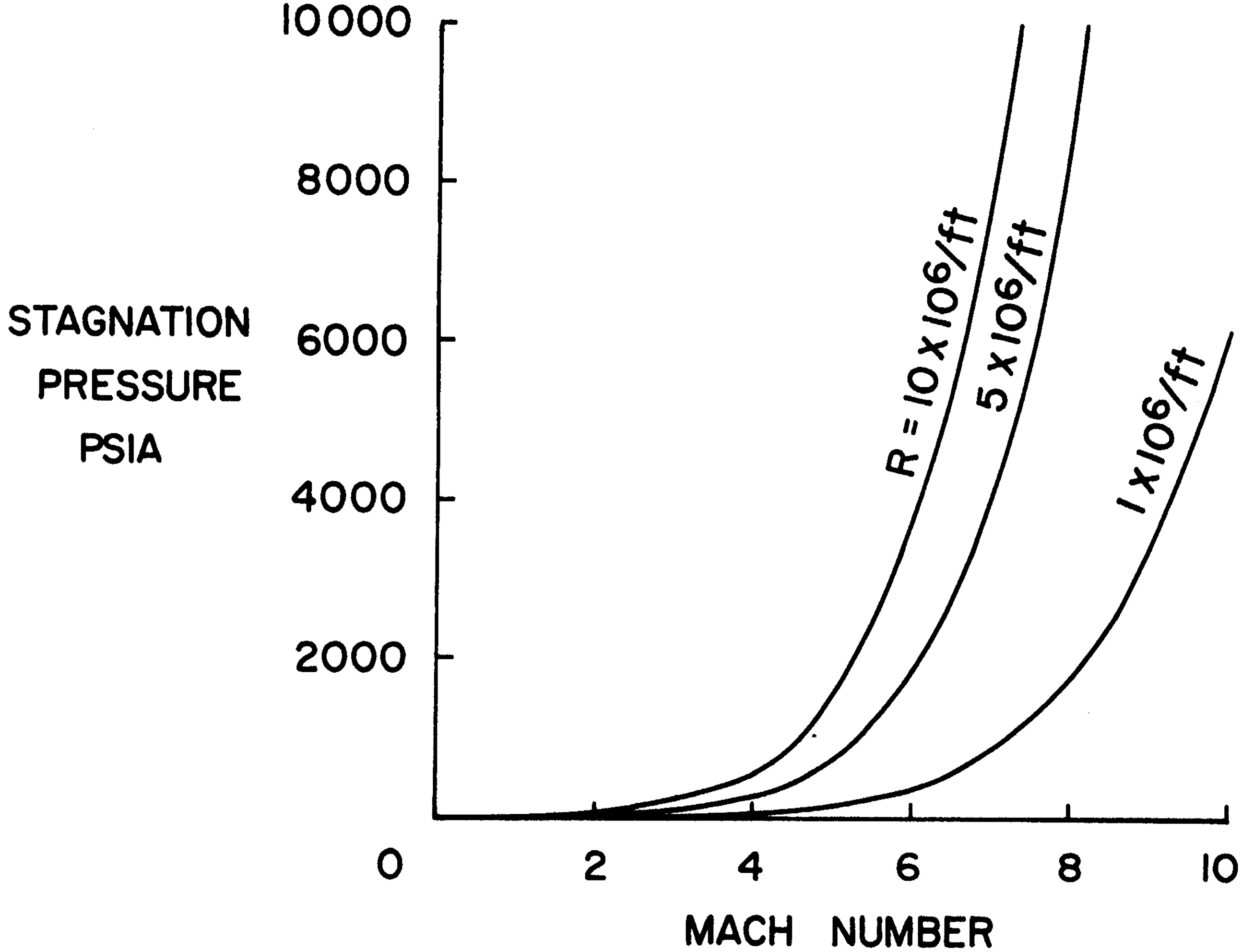

Figure 5

HEATED PLATE MODEL

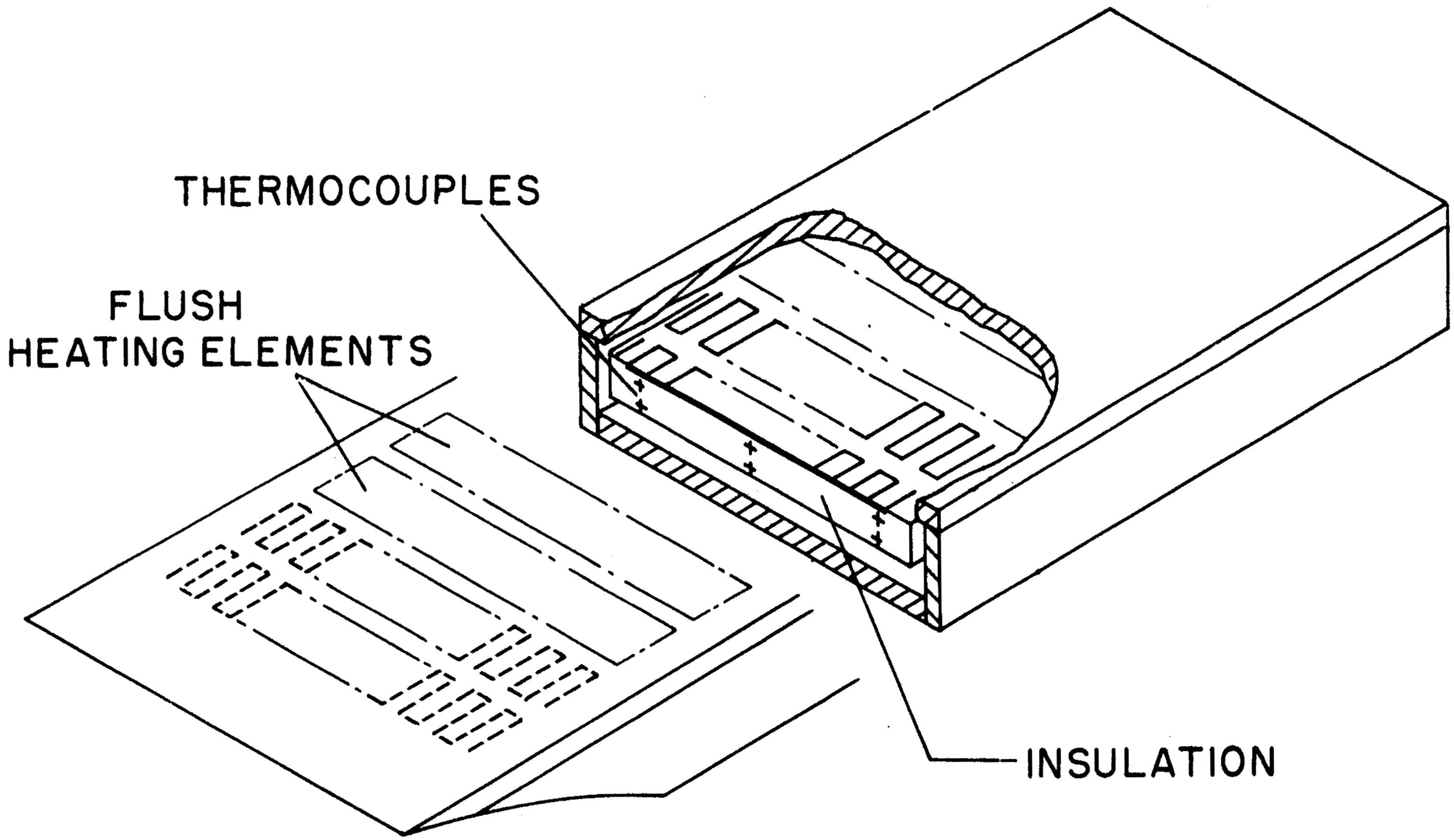

Figure 6(a)

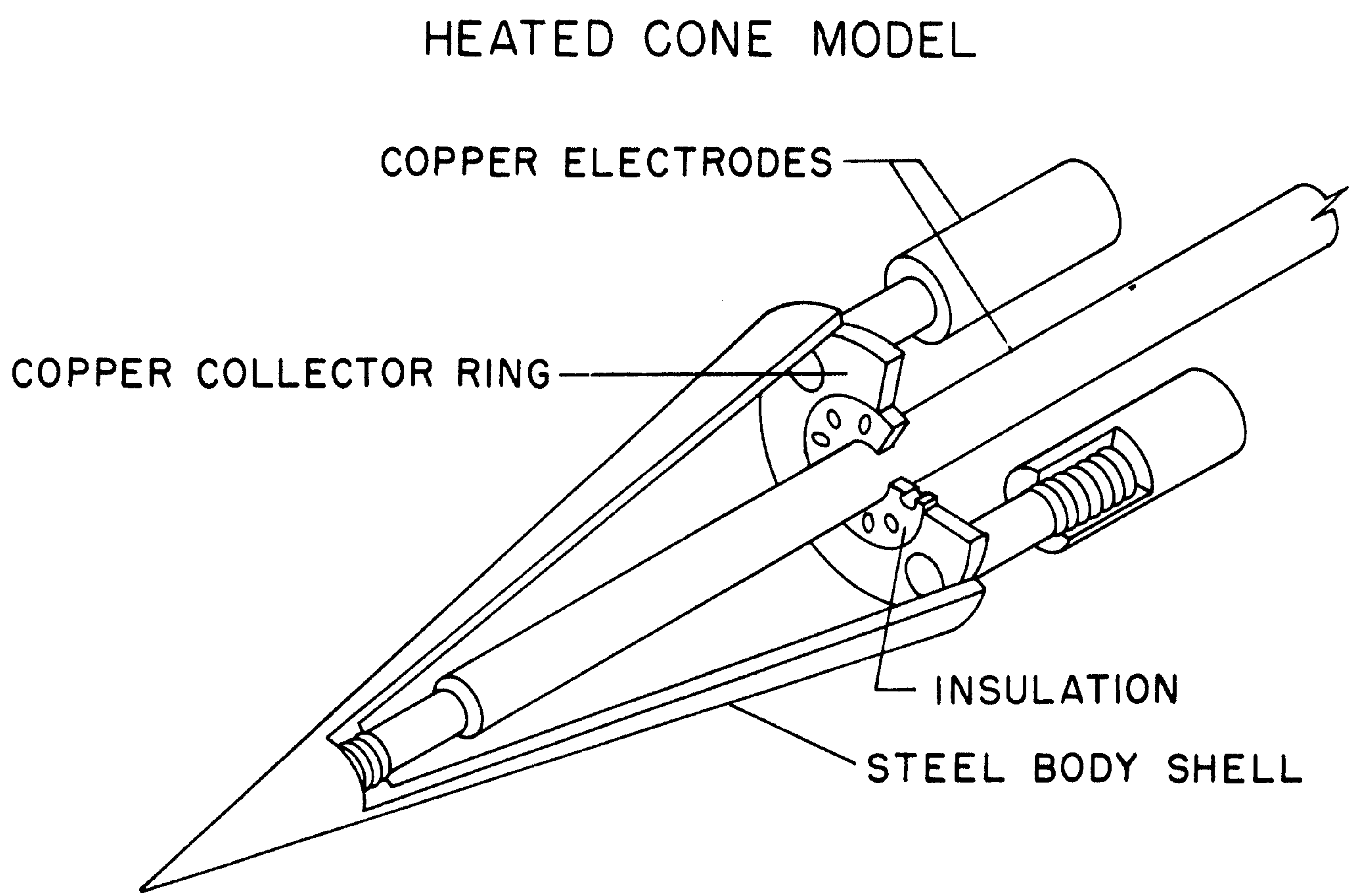
HEATED CONE MODEL
COPPER ELECTRODES
COPPER COLLECTOR RING
INSULATION
STEEL BODY SHELL

Figure 6(b)

INTERFEROGRAM OF MODEL IN FREE-FLIGHT

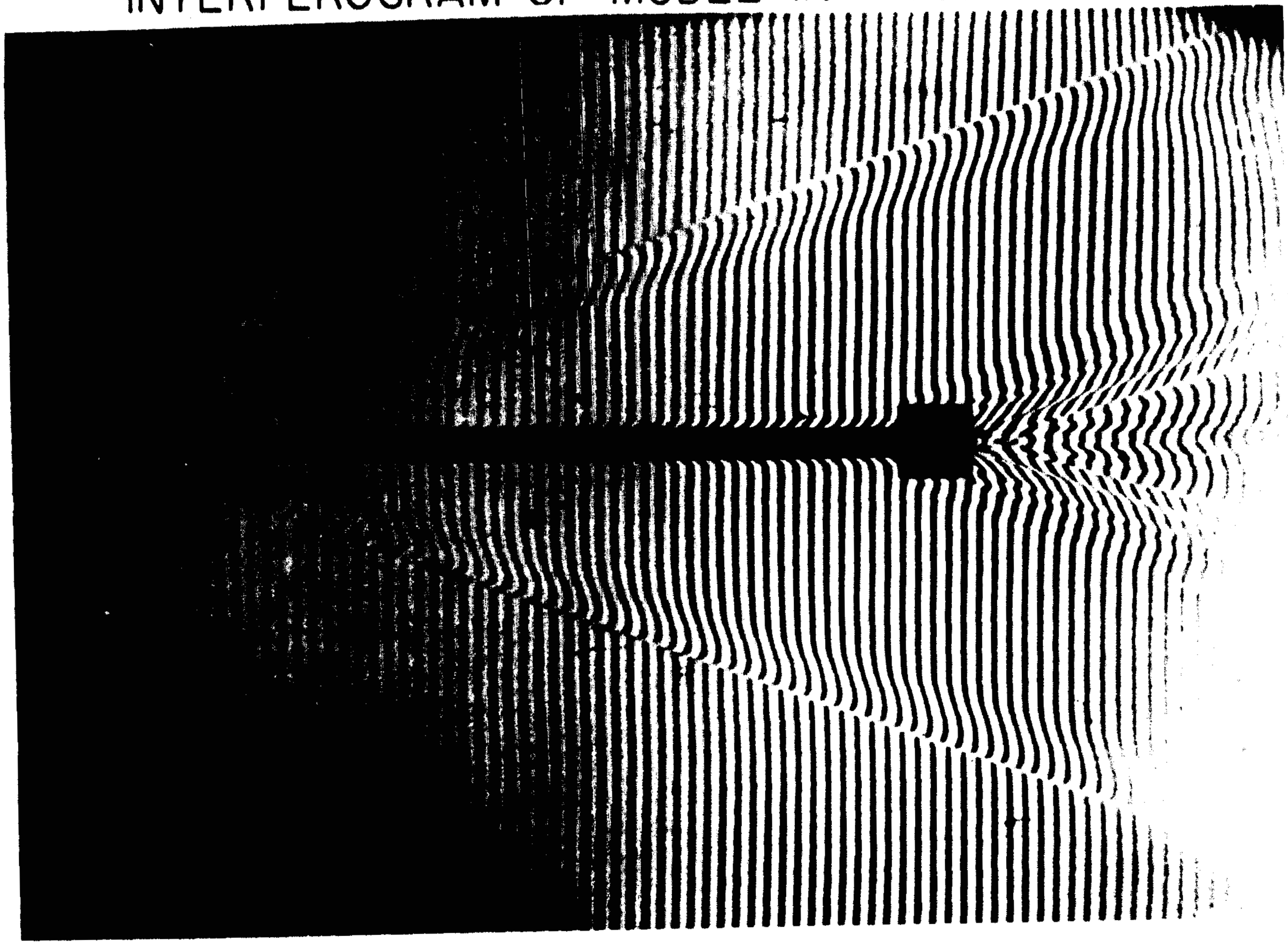

Figure 7

SHADOWGRAPH OF A SIMILAR MODEL

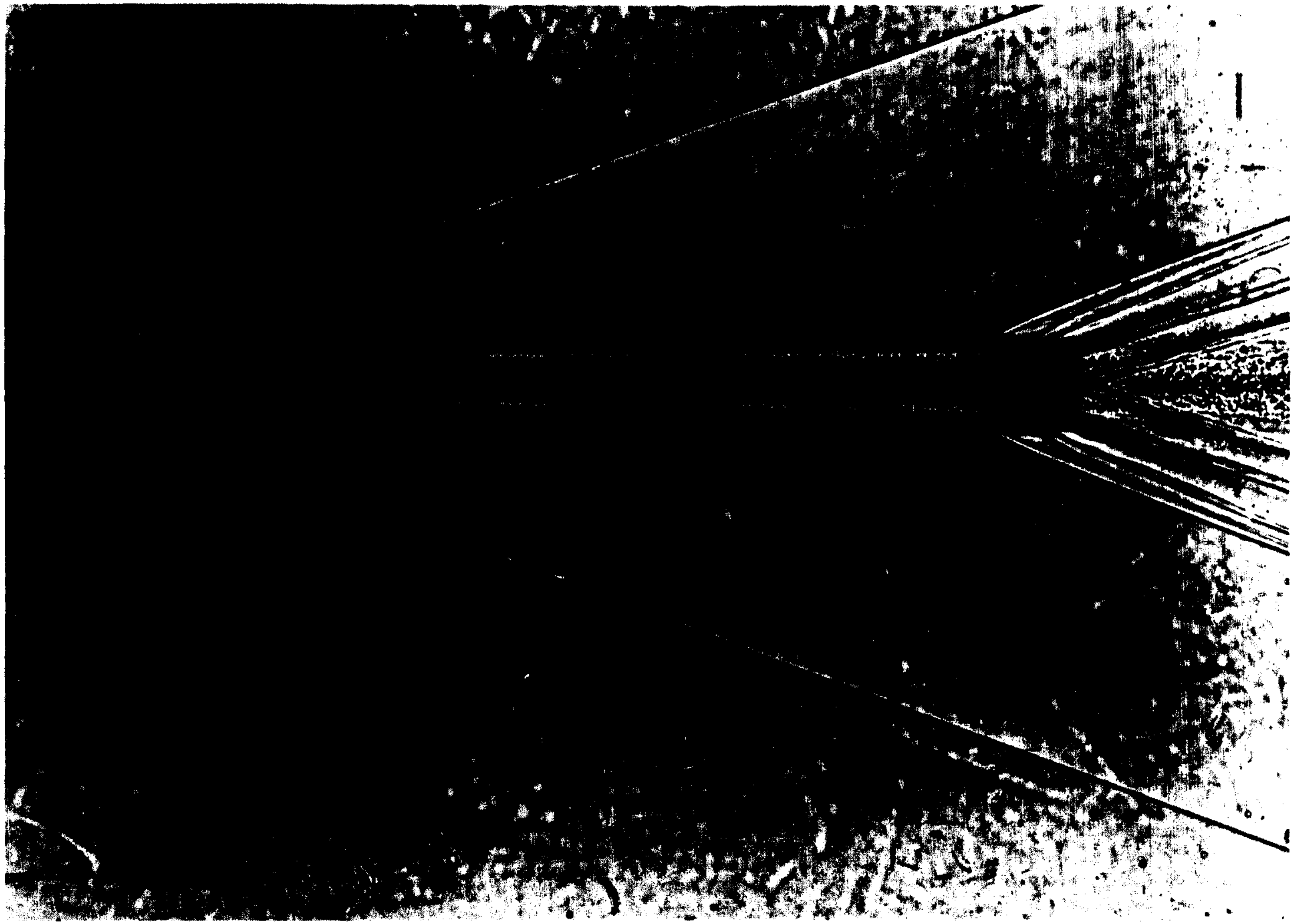

Figure 8

TEMPERATURE PROFILES MEASURED WITH INTERFEROMETER

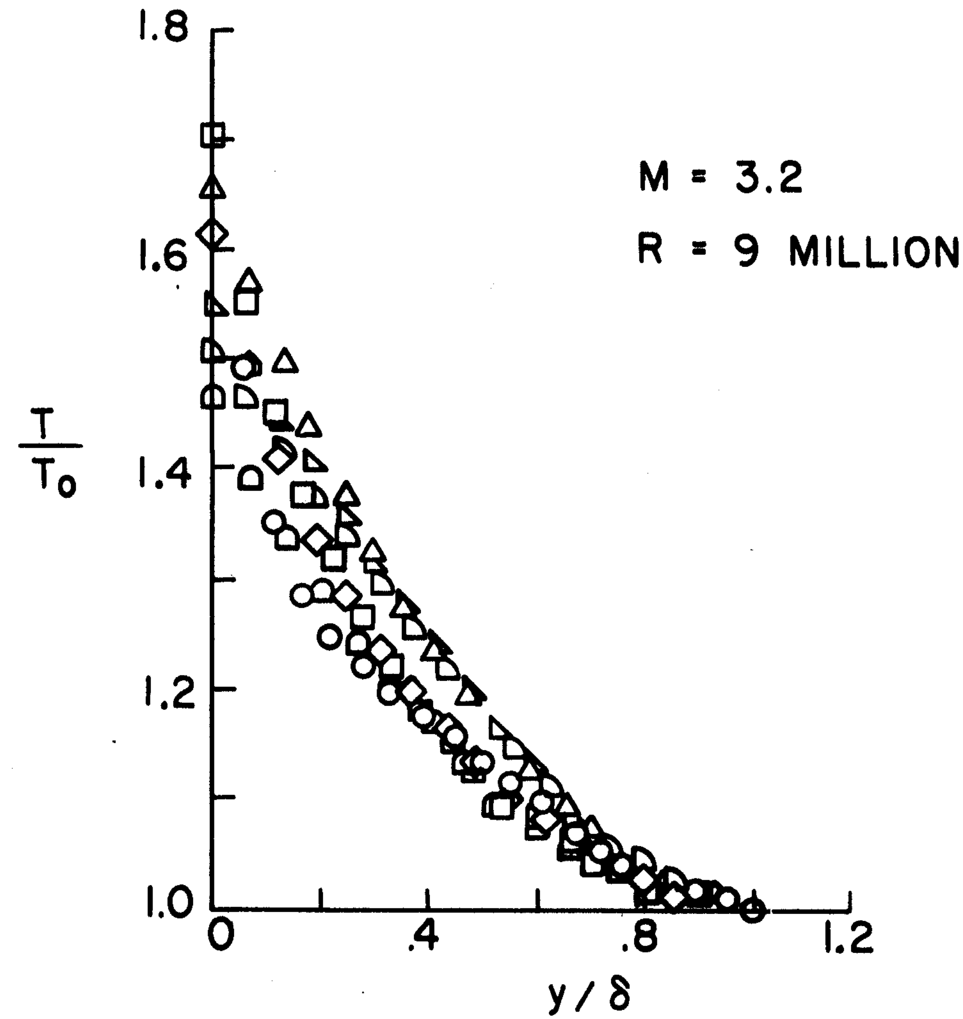

Figure 9